Payload Isolation System for Launch Vehicles

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Payload isolation system for launch vehicles

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ABSTRACT

A spacecraft is subjected to very large dynamic forces from its launch vehicle during its ascent into orbit. These large forces place stringent design requirements on the spacecraft and its components to assure that the trip to orbit will be survived. The severe launch environment accounts for much of the expense of designing, qualifying, and testing satellite components. Reduction of launch loads would allow more sensitive equipment to be included in missions, reduce risk of equipment or component failure, and possibly allow the mass of the spacecraft bus to be reduced. These benefits apply to military as well as commercial satellites. This paper reports the design and testing of a prototype whole-spacecraft isolation system which will replace current payload attach fittings, is passive-only in nature, and provides lateral isolation to a spacecraft which is mounted on it. This isolation system is being designed for a medium launch vehicle and a 6500 lb spacecraft, but the isolation technology is applicable to practically all launch vehicles and spacecraft, small and large. The feasibility of such a system on a small launch vehicle has been demonstrated with a system-level analysis which shows great improvements. The isolator significantly reduces the launch loads seen by the spacecraft. Follow-on contracts will produce isolating payload attach fittings for commercial and government launches.

Keywords: launch vehicle, vibration, isolation, attenuation, spacecraft, spacecraft isolation, payload isolation, launch loads

1. Introduction

One of the most severe environments that a spacecraft will be subjected to during its lifetime will occur during launch. This paper summarizes the results and status of a research effort in the area of spacecraft isolation from the launch vibration environment. The object of this effort was to reduce the launch-induced structure-borne dynamic acceleration of the spacecraft by insertion of a vibration isolation device. The term launch loads refers to all loads from liftoff through final engine shutdown at orbit insertion. Isolation issues involving the use of passive elements and launch vehicle system-level requirements will be discussed.

Phillips Laboratory (PL) Space Vehicle Technologies Division of the Space Technology Directorate has been monitoring the development of whole-spacecraft isolation. The result of this effort has been an isolation design methodology developed from a system-level point of view. This methodology, along with models and simulations will be used to develop new spacecraft payload attach fitting (PAF) designs which incorporate vibration isolation capability. PL is developing the technology for whole-spacecraft isolation in two phases. The first phase, discussed in this paper, is the development of passive isolation designs.[1,2] The second phase will add active control elements to develop a hybrid passive/active vibration isolation system.[3] These whole-spacecraft isolation technologies could be used to great advantage in many future launches of both government and commercial spacecraft such as the proposed constellation of satellites necessary to form global telecommunication networks.

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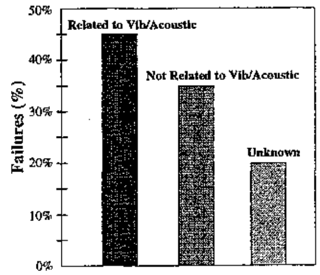


Figure 1. Causes of space flight malfunctions

2. The Need For Isolation

The deployment of a spacecraft into its final orbit configuration is a highly-complex operation. During the ascent of the launch vehicle, the spacecraft is subjected to many different quasi-static and dynamic loads which vary throughout the launch. These loads change due to environmental effects such as wind gusts and buffeting, discrete events such as motor ignitions and cutoffs, and also due to changing structural dynamics caused by fuel depletion and stage jettisons. These transient loads can have a detrimental impact on the launch survival and life cycle performance of the spacecraft. Undeniably, the load environment that spacecraft endures during launch far exceeds that encountered during on-orbit operations.

Launch dynamics are a major design driver in the structural design of a spacecraft. The vibrations that occur in a spacecraft during launch are both structure-borne and acoustic in nature. It is well established that a significant number of spacecraft malfunctions occur during launch, and that they are often due to vibro-acoustic loads. A NASA study,[4] shown in Figure 1, estimates that 45 percent of all first-day spacecraft failures and malfunctions are known to be attributed to damage caused by vibrations. While the study is over twenty years old, the problem has changed little.

Payload attach fittings are used to provide an interface between the launch vehicle (LV) and spacecraft. Typical PAFs are designed to be very stiff and subsequently they provide an efficient transmission path for both dynamic and quasi-static launch loads. The traditional approach to spacecraft design against launch vibration has been through structural stiffening or component isolation. However, this approach is costly, time consuming, adds weight, and can lead to other liabilities once the spacecraft is in orbit. Current PAFs do not provide isolation from launch loads except on a case-by-case basis. Implementing an isolation system into the PAF is the logical place for a payload isolator. However, whole-spacecraft isolation is a substantial change in the dynamic properties of the combined system and is bound to have side effects which must be addressed. Critical to the acceptance for flight is that an isolation system must not introduce intractable new problems into either the product or process. First flight of any whole-spacecraft isolator will occur only when both the LV and spacecraft contractors are satisfied that, at worst, a failure of the isolator will impose vibration on the spacecraft no worse than that which would occur with a standard PAF.

Reduced vibration environments for future spacecraft can have a direct impact on the overall cost of spacecraft design, testing, and operation. Several subsystems, such as solar arrays and other flexible structures, can be made lighter and use less expensive materials, resulting in both a mass and production cost savings. This also allows a larger percentage of the payload weight to be dedicated to scientific equipment. A whole-spacecraft isolation system is envisioned to replace the traditional PAFs used to physically attach a spacecraft to a LV as shown in Figure 2. The implementation of this technology will directly effect the following: 1) greater survivability at launch; 2) a reduction of loads seen by the spacecraft; 3) a minimization of dynamic-related spacecraft failures; 4) a reduction of cost, size, and weight of some spacecraft; 5) a lowering of certain test requirements; 6) the allowance for tuning of the isolator instead of spacecraft requalification; and 7) a reduction of the number of analysis load cycles.

In the course of a spacecraft development program, updated coupled-loads analyses often result in increased launch loads which necessitate unforeseen spacecraft design changes. Consequently, the spacecraft design organization is faced with unplanned hardware redesigns, schedule slips, and cost over-runs. Reduction of dynamic launch loads seen by the spacecraft will minimize spacecraft redesign, reduce risk, reduce spacecraft development time, reduce costs, eliminate many vibration-related failures, and increase reliability.

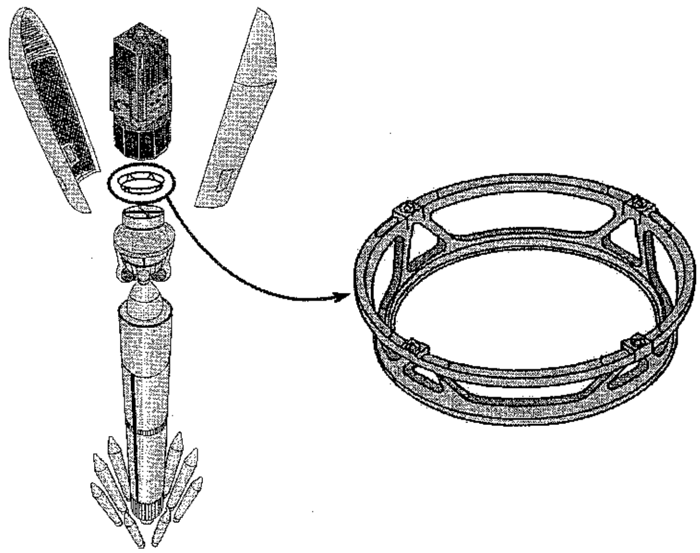


Figure 2. Medium launch vehicle and payload attach fitting

3. Isolation Design Methodology

Vibration isolation is a technique used to reduce vibration of a structure by altering the frequency content of the forces that act on that structure. Isolation of a whole spacecraft from a launch vehicle requires a unique design methodology. Figure 3 shows two connected structures being acted upon by external forces. Classic isolation design assumes that structure 2 is rigid with respect to structure 1 and only the dynamics of structure 1 must be considered in the design process. This is not at all true for whole-spacecraft isolation design. The spacecraft (structure 1) and the launch vehicle (structure 2) are both considered to be very flexible structures and the dynamics of one has significant influence on the other.

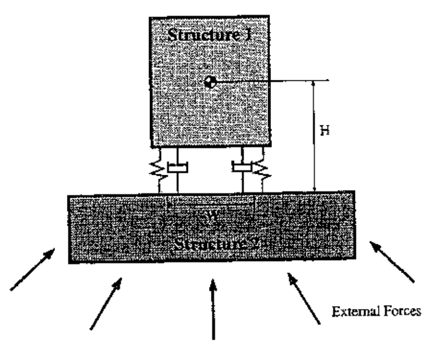


Figure 3. Two connected structures

Historically, the connection between the spacecraft and the launch vehicle has been made with a very stiff payload attach fitting. This is generally considered to be a "hard mount" and is extremely efficient at transmitting all structure-borne forces from the launch vehicle to the spacecraft over a very wide frequency band. A whole-spacecraft isolation system replaces this hard mount with a soft mount which filters out a great deal of the frequency spectrum of the forces from the LV.

Most spacecraft are cantilevered to their launch vehicle, being attached only at their base. Whole-spacecraft isolation is a challenging problem because spacecraft typically have a very large ratio of center-of-gravity height, H, to attachment width, W (Figure 3). Reduction of the axial attachment stiffness will introduce low-frequency spacecraft rocking (pitch & yaw) modes and large lateral displacements at the upper end of the spacecraft. This is generally undesirable because it may cause guidance control system instabilities or the spacecraft may hit the fairing. However, these problems may be avoided through innovative isolator hardware design.

Launch vehicles often have closely-spaced flexible modes with frequencies starting as low as 1 Hz and spacecraft may have modes with frequencies starting as low as 6 Hz. Isolation of a 6 Hz spacecraft from a 1 Hz LV necessitates a unique design methodology which deviates from classic isolation system design. More than ever, it is necessary to have a clear understanding of the isolation objectives and the design constraints present. Whole spacecraft isolation systems must be designed from a system-level point of view, accounting for the coupled dynamics of two very flexible bodies which will now be connected with a flexible interface as opposed to a hard mount. Indeed, the challenge is to determine exactly where to insert the new dynamics introduced by the isolation system within the sea of structural dynamics already present. The following sections discuss the methodology which was used to develop a whole-spacecraft isolation system for a medium launch vehicle.

3.1. Isolation objectives

The specific objectives for the design of this isolation system were the following:

- Isolate the spacecraft, as a whole, from the launch vehicle. Individual components of spacecraft have been isolated and flown, but a whole spacecraft has never been isolated.
- Provide lateral isolation only. It was decided to only reduce lateral accelerations in this program. Axial isolation, while feasible, was deemed beyond the scope of this program. This objective is tied closely to the design constraints, discussed later.
- Provide a 2:1 broadband RMS reduction in accelerations in the 25 Hz to 35 Hz range. Many spacecraft have secondary structures such as solar arrays, antennas, etc. with modes in this range; these modes will not be excited as much if isolation is designed in this range.
- Reduce accelerations on spacecraft secondary structure. Primary structure of spacecraft is usually designed to meet quasi-static loads and does not, in itself, generally require dynamic load reduction.

3.2. Design constraints

There are many design constraints which pertain to whole-spacecraft isolation. Some of the typical constraints are weight, volume, and strength. However, the two most critical design constraints are:

1. Must not introduce structural modes below 6 Hz. This constraint is related to the vehicle guidance, navigation, and control systems. Structural modes below 6 Hz encroach on the controller bandwidth and may cause flight instabilities.
2. Must not increase the rattle displacement (payload-to-fairing displacement) by more than one inch. Insertion of a whole-spacecraft isolator will introduce compliance between the LV and the spacecraft. This compliance must not significantly increase the rattle displacements which could cause the spacecraft to hit the fairing during launch.

3.3. System-level analysis

Realistic and thorough system-level mathematical models are required to properly design and analyze the system-level benefits of whole-spacecraft isolation. The correct approach to designing isolation for a launch vehicle and spacecraft system is to use finite element models of all parts of the system. This allows accurate simulation of the structural dynamics of the non-isolated system and subsequently provides a tool for simulation of various isolation hardware designs.

The launch vehicle changes significantly during its ascent, due to fuel depletion and stage jettisons. Therefore, many LV models and associated loads would be required to fully analyze any isolator design. For the purpose of designing this isolation system, two flight events were selected: liftoff and pre main-engine cutoff (preMECO). Separate finite element models were obtained for a generic medium launch vehicle, representing these two distinct periods of launch. The first is a liftoff model, matching the vehicle as it sits on the launch pad. This model was obtained in matrix form only, with 185 physical degrees of freedom (DOF) and 49 modal DOF, for a total of 234 DOF. The second finite element launch vehicle model represents the preMECO period of flight. It was also obtained in matrix form only, with 12 physical DOF and 139 modal DOF, for a total of 151 DOF.

A realistic model was obtained of a NASA spacecraft which weighs 6500 lb and is 16 feet in height. This model originally consisted of a bus structure and 14 substructured equipment items totaling more than 20,000 DOF. This was all combined into one modal-reduced spacecraft substructure having 138 physical DOFs and 148 modal DOFs for a total of 286 DOF. This model is representative of the complicated high-modal-density dynamics present in a typical spacecraft, and was therefore very useful in the isolation design.

The substructuring facility of UAI/NASTRAN was used to simplify the system-level analyses. The launch vehicle models were each stored in the database as separate substructures, as was the spacecraft. The only changing component in each analysis was the PAF, which was also substructured. The assembly of a system-level model involved combining the spacecraft substructure, the current PAF iteration substructure, and the desired launch vehicle substructure into a single system. Then this system was analyzed using either frequency response or transient response solutions. This process is illustrated in Figure 4. Direct solutions were quite feasible, as opposed to modal solutions, because the substructuring significantly reduced the solution matrix sizes.

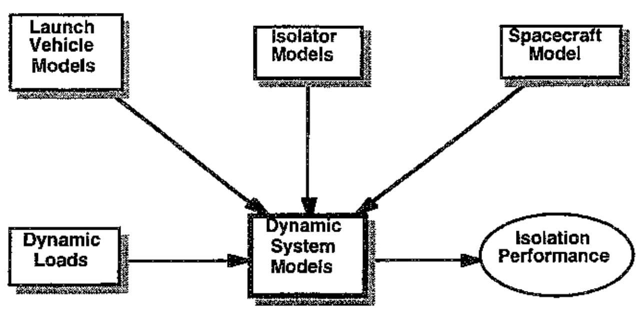


Figure 4. System-level dynamic analysis

Selection of an isolation system design was an iterative procedure, in which each concept was analyzed in a dynamic system model to assess its performance characteristics. Since the main isolation target was the preMECO stage of flight, this model and preMECO loads were used for preliminary evaluation of isolation designs. Initially, the isolation system was modeled as a set of springs and dampers between the launch vehicle and the spacecraft. This allowed rapid trade studies to be performed with several variables such as lateral stiffness, axial stiffness, and damping. Full finite element models of the isolating PAF (IPAF) were used once the design progressed.

The most useful analysis method for the isolation trade studies was frequency response analysis. Using this method, transfer functions were generated between main-engine force inputs and spacecraft acceleration outputs. Comparison of these transfer functions between non-isolating and isolating PAFs provided considerable insight into the isolation performance. Figure 5 shows the isolation performance for the final isolating PAF design in this program. This figure shows that the acceleration response will be greatly reduced in the 25 Hz to 35 Hz frequency range. The amount of broadband attenuation may be indicated by a single number called the "RMS ratio". This is simply the ratio of the RMS of the isolated acceleration PSD to the RMS of the non-isolated acceleration PSD when subjected to uniform white noise input. For the final design, the RMS ratio is 0.39 over the 0 Hz to 40 Hz frequency band, and it is 0.17 over the 20 Hz to 40 Hz frequency band. This exceeds the program goal of an RMS ratio of 0.50.

A thorough coupled-loads analysis was done to evaluate the final design for many other load cases. Table 1 shows the reductions in accelerations due to the isolator and the RMS ratios. These acceleration values were the peak values from all load cases analyzed. The IPAF has reduced the peak lateral accelerations by as much as 46%. The system-level analysis shows that the isolating payload attach fitting provides excellent lateral vibration isolation for the spacecraft.

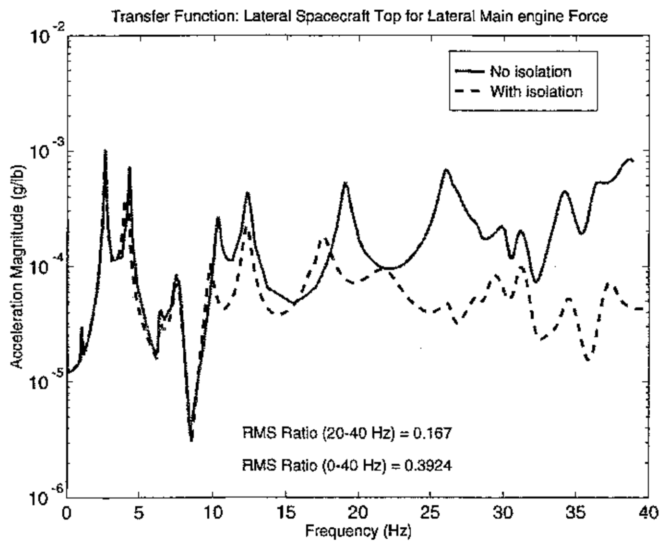
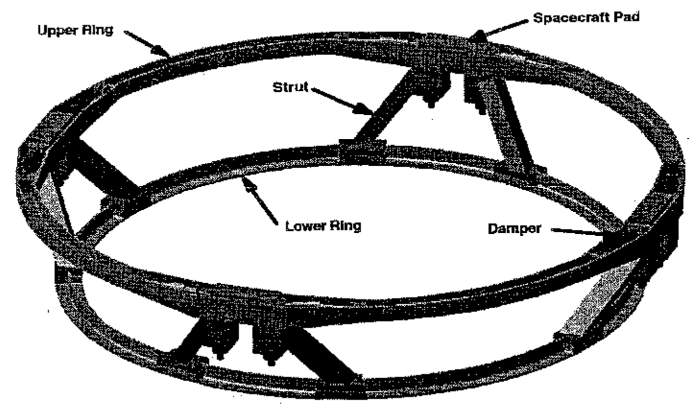


Figure 5. Transfer function showing lateral isolation

Location	Dir*	Max Overall† Acceleration Change	PreMECO Acceleration Change	RMS Ratio§
Top of spacecraft	X	-30%	-33%	0.39
	Y	-31%	-33%	0.37
	Z	-9%	-69%	0.77
Component on top of spacecraft	X	-41%	-46%	0.37
	Y	-29%	-30%	0.38
	Z	-14%	-70%	0.42

* X and Y are lateral directions; Z is axial
†"Max Overall" is the max of liftoff, transonic, and max Q
§"RMS Ratio" is the isolated RMS acceleration divided by the non-isolated RMS acceleration (0-40 Hz)

Table 1. Summary of isolation performance



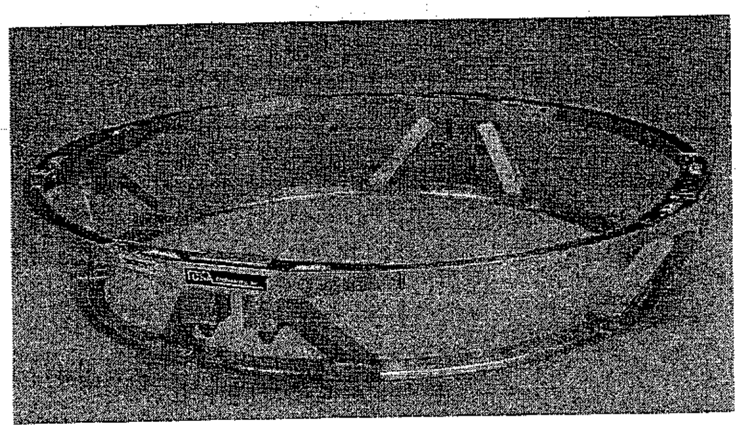


Figure 6. PIP isolating payload attach fitting: solid model and full-scale hardware (69" diameter)

3.4. Component-level analysis

The isolation system consists of both stiffness components and damping components. The system-level analysis was used to arrive at the optimum values for stiffness and damping of this isolator. Then, using these requirements, the isolator stiffness and damping elements were designed. This process consisted of both hardware design and component-level analyses, using detailed finite element models, to size and verify the design.

4. Hardware Design & Fabrication

One purpose for building hardware on this program was so that it could be tested and the resulting data be used for tuning the mathematical models. High confidence in the isolating PAF mathematical model gives high confidence in the full system-level coupled-loads analysis results.

The final design for the isolating PAF structure and its full-scale hardware implementation are shown in Figure 6. This design is intended to be a "slip-in" replacement for the existing hard-mount PAF. Care was taken to match the same basic dimensions and bolt patterns. The lower ring bolts to the upper stage of the launch vehicle. The spacecraft bolts at four locations to the spacecraft pads. The load path from the spacecraft to the launch vehicle goes through the spacecraft pads into a flexure system (not shown), then into the upper end of the struts, then down to the lower ring, and finally into the launch vehicle. Space is left between the upper ends of the strut pairs to accommodate a pyrotechnic nut at each spacecraft mounting location.

The original hard-mount PAF, which has flown many times, is fabricated from a monolithic piece of aluminum. The resulting structure has no welds and is extremely costly to manufacture. To avoid prohibitive costs in this program, the full-scale hardware for the isolating PAF was made from several pieces welded together. Both the lower ring and the upper ring were made from eight machined pieces welded together. The struts were bolted to the upper and lower rings. This was a perfectly reasonable approach for building a non-flight version of this isolating PAF. A flight version of the isolating PAF would not have any welds or bolted strut joints.

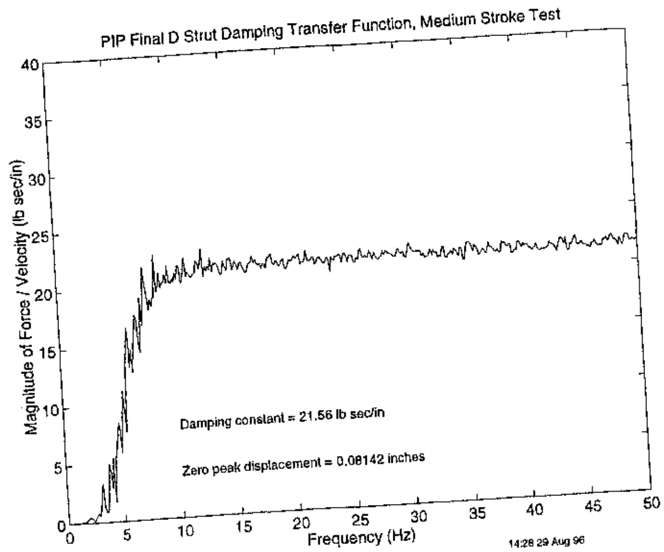


Figure 7. Results from medium-stroke test of PIP D-strut

5. Hardware Tests

Several tests were performed to measure the stiffness and damping of the hardware for the purpose of test-verifying the mathematical models.

The damping element, which is a version of Honeywell's D-strut, behaves like a viscous dashpot and was tested to measure its damping constant. Direct complex stiffness testing resulted in force/velocity transfer functions such as that shown in Figure 7. This shows the magnitude of the force/velocity transfer function for a damping strut which was subjected to a medium-stroke and 50 Hz bandwidth test. The measured damping constant is independent of frequency, for all practical purposes, and meets the requirements for the system.

The isolator stiffness elements, which consist of a system of flexures, were tested for both their axial and lateral stiffness values. The isolation performance, at the system level, is crucially dependent on the stiffness of these flexures. Table 2 shows a comparison of the stiffnesses from both the test and the finite element model of the flexures. This shows excellent correlation between the model and the test data, indicating that the flexures behave in test exactly as they were designed to.

Stiffness Direction	Test Stiffness (lb/in)	Model Stiffness (lb/in)	Difference
Lateral	15,540	15,580	+0.3%
Axial, Tension	1,670,000	1,660,000	-0.6%
Axial, Compression	2,018,000	2,011,000	-0.3%

Table 2. Comparison of the stiffnesses of the flexure system

A modal test was performed on the complete assembled structure to verify that the finite element model is accurate and correctly predicts the behavior of the isolating payload attach fitting. This test was designed to simply extract the first few modes of the structure for use in tuning and validating the model. The measures of comparison between the test and analysis results were a frequency comparison and a mode shape cross-orthogonality matrix. The frequency comparison is shown in Table 3. It can be seen that the frequencies for the first several modes match within about 5%, indicating good correlation.

Frequency (Hz)		Difference
Test	Analysis	
20.65	19.56	-5.3%
33.75	32.21	-4.6%
37.38	37.78	1.1%
77.06	81.78	6.1%
83.50	75.70	-9.3%

Table 3. Frequency comparison between test and analysis

The mode shapes were compared by calculating the cross-orthogonality matrix between test and analysis mode shapes. An analytical reduced mass matrix was calculated, using Guyan reduction in NASTRAN, and was used in the cross-orthogonality calculation. All data were imported into MATLAB for this process. The cross-orthogonality matrix is shown in Table 4. This matrix indicates that there is excellent correlation between the finite element model and the hardware.

		Analysis				
	Freq (Hz)	*19.56*	*32.21*	*37.78*	*81.78*	*75.70*
Test	*20.65*	**0.993**	0.012	0.085	0.000	0.023
	33.75	0.006	**0.995**	0.030	0.078	0.003
	37.38	0.083	0.046	**0.991**	0.002	0.073
	77.06	0.015	0.048	0.014	**0.970**	0.020
	83.50	0.107	0.007	0.202	0.003	**0.844**

Table 4. Modal test & analysis cross-orthogonality matrix

6. Small Spacecraft / Small Launch Vehicle Isolation

While the preceding places emphasis on isolation of a large 6500 lb spacecraft from a medium launch vehicle, it must be clarified that similar isolation methods may also be applied to a smaller class of problems. Whole-spacecraft isolation of a small spacecraft (less than 1500 lb) from the launch environment of a small launch vehicle is an extremely feasible task. Indeed, the isolation problem becomes easier because the ratio of center-of-gravity height to attachment width (discussed previously) is generally smaller. This minimizes collateral problems of low-frequency rocking and payload-to-fairing clearance reduction. A study was performed on a 600 lb spacecraft and a small launch vehicle to assess the feasibility of launch isolation. Figures 8 and 9 show the significant reductions in spacecraft acceleration loads that may be achieved in the lateral and axial directions, respectively. Isolation designs are currently in progress for this class of problem and are considered very promising.

7. Summary

There is a need to reduce launch loads on spacecraft so that spacecraft and their instruments can be designed with more concentration on orbital performance rather than launch survival. A softer ride to orbit will allow more sensitive equipment to be included in missions, reduce risk of equipment or component failure, and possibly allow the mass of the spacecraft bus to be reduced. These benefits apply to military as well as commercial satellites.

The approach taken in this work was to incorporate an isolation system into the payload attach fitting, which is the structure that connects the spacecraft to the launch vehicle. The isolation system was to provide lateral isolation in the 25 - 35 Hz range, an important dynamic range for secondary equipment.

Whole-spacecraft isolation is a challenging problem requiring a great deal of system-level and detail design engineering. Using realistic models of a launch vehicle and spacecraft, coupled-loads analyses were performed for several

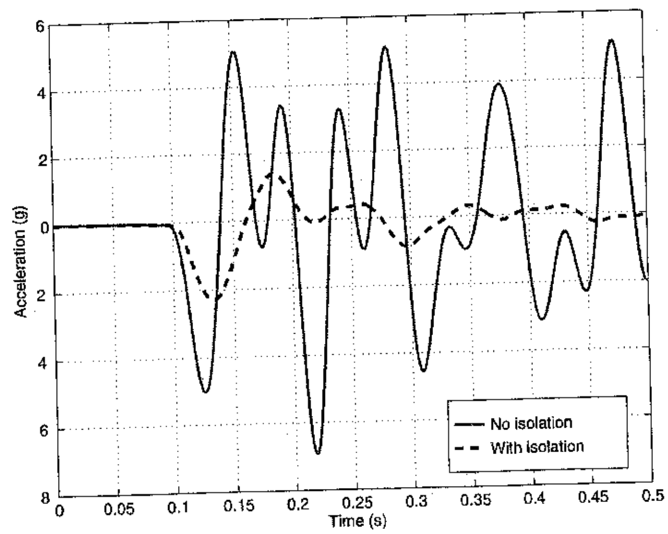


Figure 8. Lateral isolation performance for small spacecraft / small launch vehicle

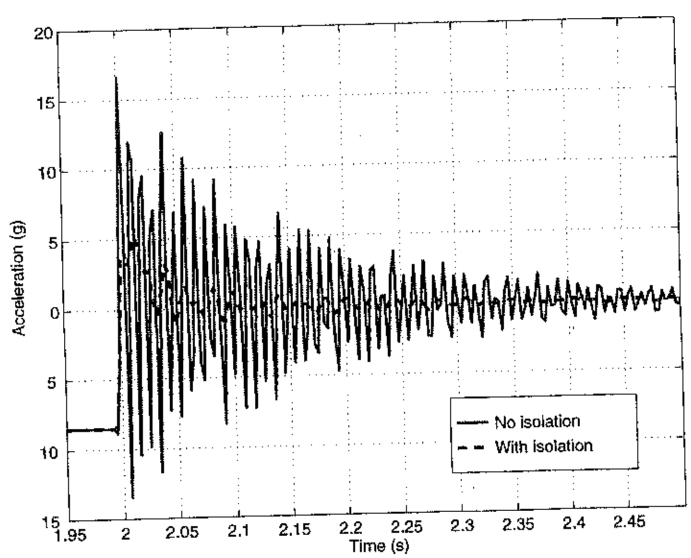


Figure 9. Axial isolation performance for small spacecraft / small launch vehicle

flight events to determine the optimum isolation parameters. Once these parameters were determined, detailed design analysis was used to develop hardware that would produce the desired results. Full-scale prototype hardware (69 inches in diameter) was fabricated and tested to verify the analytical models. The isolating payload attach fitting was a one-for-one replacement for the original. At the conclusion of the design phase, complete (all cases) coupled-loads analyses were also performed to verify the performance of the isolation system.

Additionally, isolation of small spacecraft from small launch vehicles is seen as a very tractable problem which may provide significantly softer rides on these vehicles, which typically have solid rocket motors.

This work brings technology to the launch community which may significantly reduce launch vibration problems and reduce risk of spacecraft component failure.

8. Acknowledgments

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